USE OF THE SDO POINTING CONTROLLERS FOR INSTRUMENT CALIBRATION MANEUVERS

Melissa F. Vess, Scott R. Starin, and Wendy M. Morgenstern
NASA Goddard Space Flight Center, Greenbelt, MD 20771

ABSTRACT

During the science phase of the Solar Dynamics Observatory mission, the three science instruments require periodic instrument calibration maneuvers with a frequency of up to once per month. The command sequences for these maneuvers vary in length from a handful of steps to over 200 steps, and individual steps vary in size from 5 arcsec per step to 22.5 degrees per step. Early in the calibration maneuver development, it was determined that the original attitude sensor complement could not meet the knowledge requirements for the instrument calibration maneuvers in the event of a sensor failure. Because the mission must be single fault tolerant, an attitude determination trade study was undertaken to determine the impact of adding an additional attitude sensor versus developing alternative, potentially complex, methods of performing the maneuvers in the event of a sensor failure.

To limit the impact to the science data capture budget, these instrument calibration maneuvers must be performed as quickly as possible while maintaining the tight pointing and knowledge required to obtain valid data during the calibration. To this end, the decision was made to adapt a linear pointing controller by adjusting gains and adding an attitude limiter so that it would be able to slew quickly and still achieve steady pointing once on target. During the analysis of this controller, questions arose about the stability of the controller during slewing maneuvers due to the combination of the integral gain, attitude limit, and actuator saturation. Analysis was performed and a method for disabling the integral action while slewing was incorporated to ensure stability. A high fidelity simulation is used to simulate the various instrument calibration maneuvers.

INTRODUCTION

The Solar Dynamics Observatory (SDO), scheduled to launch in 2008, will carry a suite of three instruments — the Atmospheric Imaging Assembly (AIA), the Helioseismic & Magnetic Imager (HMI), and the Extreme Ultraviolet (EUV) Variability Experiment (EVE) — designed to provide observations leading to a more complete understanding of the solar dynamics that drive variability in the Earth's environment. The vehicle's geosynchronous orbit allows for uninterrupted, high-rate downlink of the science data. The observatory maintains a fixed attitude relative to the Sun, allowing the instruments to collect a steady stream of solar images.

AIA, built by Lockheed Martin Solar and Astrophysics Laboratory (LMSAL), seeks to investigate the evolution of the Sun's magnetic field through the use of coronal images. AIA also includes four guide telescopes (GT) that are used to drive the instrument's Image Stabilization System (ISS), which prevents high frequency jitter from blurring the image data. The GTs are high precision Sun sensors with a fine pointing range of approximately ± 95 arcsec. The data from one of the guide telescopes, called the controlling guide telescope (CGT), is used by the on-board attitude control system (ACS) during Science mode to measure attitude errors relative to the Sun center. HMI, built jointly by LMSAL and Stanford University, is designed to use observations of polarized light to measure the magnetic field and velocity of the solar photosphere. EVE, built by the Laboratory for Atmospherics and Space Physics (LASP) at the University of Colorado, seeks to understand the highly variable solar EUV electromagnetic radiation and its impacts on the geospace environment. More information about the SDO science instruments can be found through the SDO website.[1]

SDO's science, in particular HMI's helioseismology, requires twenty-two individual 72-day periods over five years to meet the full science data capture objectives. SDO meets this requirement by providing near-continuous Sun-pointing observations with a few interruptions for necessary maneuvers such as orbit maintenance, momentum management and science instrument calibration. However, these interruptions must be minimized in order to successfully meet the data capture budget.

To ensure the continuing validity of the science data collected by the three instruments, once every three months the mission team will command a series of instrument calibration maneuvers sweeping the instrument boresights across the Sun in small steps. Some of these maneuvers pull the Sun out of the fine-pointing range of the GT, and therefore attitude control during these maneuvers must be performed without use of the GT data. The ACS Inertial mode must, therefore, provide the required fine pointing and accurate knowledge of the vehicle's attitude relative to the Sun center using the attitude knowledge of the ACS sensor suite. These requirements pose two separate and challenging design problems for the ACS team: ensuring adequate knowledge of the spacecraft attitude, and designing a controller that can both slew quickly to each calibration point and point accurately and stably once there.

SDO INSTRUMENT CALIBRATION MANEUVERS

SDO has a total of five different science instrument calibration maneuvers that are performed periodically throughout the mission lifetime. Four of the maneuvers point the spacecraft off of Sun center along the Y- and Z-axes of the instruments' fields-of-view. The fifth maneuver rolls the spacecraft about the body X-axis. Figure 1 shows a diagram of the spacecraft and body axes, with each of the science instruments labeled. The following section gives a brief description of each of these maneuvers.

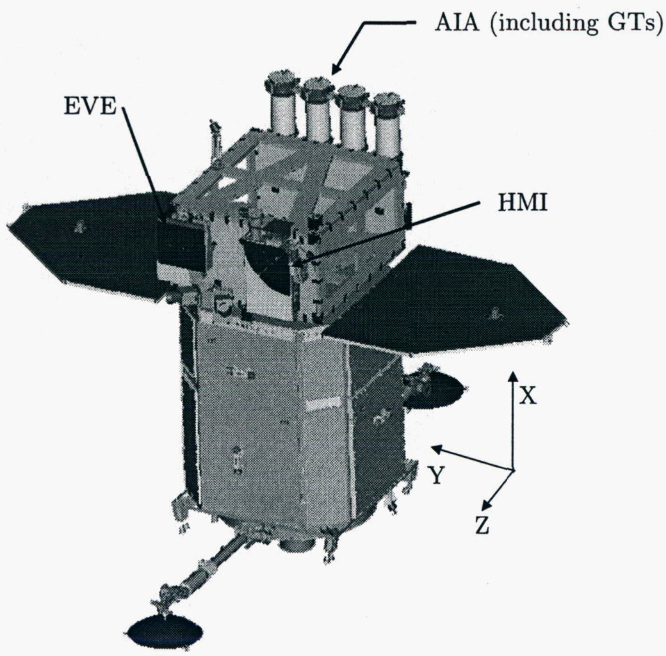

Figure 1: SDO spacecraft with instruments labeled.

EVE Cruciform Maneuver

The EVE cruciform maneuver is performed in Inertial mode four times per year. It is the longest calibration maneuver in both number of steps and maneuver duration. It starts with a 2.5-degree slew such that the Observatory is pointed off of Sun center, placing the Sun center on the positive Y-axis of the EVE field-of-view. The spacecraft then steps back towards and through Sun center in 180-arcsec steps, dwelling 30 seconds at each calibration point. This process continues until the Observatory points 2.5 degrees off Sun center along the negative Y-axis. The spacecraft then slews back to Sun center and repeats the slew-and-step procedure about the Z-axis. The 202-step maneuver pattern moves the Sun center in a cruciform along the spacecraft and instrument Y- and Z-axes.

Each 180-arcsec slew must be completed in two minutes or less, and each 2.5-degree slew must be completed in five minutes or less. Over each 30-second dwell, the ACS must limit the spacecraft angular rate to less than 6 arcsec/sec. In order for EVE to obtain valid data, the Observatory must provide a position knowledge requirement of at least ± 90 arcsec, 3σ, with respect to Sun center. Of that 90 arcsec, the ACS allocation is 35/70/70 arcsec, 3σ.

EVE Field of View Maneuver

The EVE field of view (FOV) maneuver is a 5×5 step raster scan performed in Inertial mode four times per year. The FOV size is ± 600-arcsec along the Y- and Z-axes, and each calibration step is 300 arcsec. The spacecraft dwells 60 seconds at each of the 26 calibration points.

The ACS must complete each 300-arcsec slew in two minutes or less and must maintain each dwell attitude to within ± 60 arcsec over the 60-second dwell. The Observatory must provide position knowledge of at least ± 90 arcsec, 3σ, with respect to Sun center. Of that 90 arcsec, the ACS allocation is 35/70/70 arcsec, 3σ.

HMI/AIA Off-Point Maneuver

The HMI/AIA off-point maneuver is performed in Inertial mode twice a year. It is an off-point of approximately ± 0.25 deg (about half the solar disk) along the Y- and Z-axes. There are approximately 20 steps, which vary in size from 100 to 1800 arcsec. The spacecraft dwells five minutes at each calibration position.

The ACS has two minutes to slew between points up to 300 arcsec apart and five minutes to slew between steps greater than 300 arcsec. The Observatory must provide attitude knowledge of at least ± 90 arcsec, 3σ, with respect to Sun center. Of that 90 arcsec, the ACS allocation is 35/70/70 arcsec, 3σ.

HMI/AIA Roll Calibration Maneuver

The HMI/AIA roll calibration maneuver is performed in Science mode two times per year. It consists of a 360-degree roll about the Sun line, in steps of approximately 22.5 deg, for a total of 16 steps. The spacecraft dwells 15 minutes at each calibration point, and the ACS has 10 minutes to slew between points. The Observatory must provide attitude knowledge of at least ± 90 arcsec, 3σ, with respect to Sun center. Of that 90 arcsec, the ACS allocation is 35/70/70 arcsec, 3σ.

AIA Guide Telescope Calibration Maneuvers

There are two different AIA guide telescope calibration maneuvers: a long form maneuver and a short form maneuver. Both maneuvers are performed in Science mode. The long form maneuver will be performed for each of the four GTs during instrument commissioning. The short form maneuver will be performed on a monthly basis only for the CGT.

Long Form Maneuver

The long form guide telescope calibration maneuver is intended to find and calibrate the linear range of each of the guide telescopes. The maneuver uses a cruciform pattern in which the spacecraft slews approximately ± 100–105 arcsec off of Sun center along the Y- or Z-axis and steps back towards and through

Sun center in steps of 5 to 30 arcsec, dwelling 60 seconds at each calibration point. The spacecraft has 2 minutes to slew between calibration points. The procedure is then repeated for the other GT axis.

Short Form Maneuver

The short form CGT calibration maneuver is used to periodically recalibrate the CGT and the AIA instrument stabilization system (ISS). As with the long form maneuver, the short form maneuver uses a cruciform scan pattern. The spacecraft slews approximately ±30 arcsec from its nominal science pointing position along the Y- or Z-axis and steps back towards and through nominal science pointing in steps of 5 to 30 arcsec, dwelling 1 to 5 minutes at each calibration point. The spacecraft has 2 minutes to slew between calibration points. The procedure is then repeated for the other spacecraft axis.

ATTITUDE DETERMINATION TRADE STUDY

With instrument calibration maneuvers of up to approximately nine hours in duration, the calculation of the spacecraft's inertial attitude relative to the Sun center must compensate for the apparent motion of the center of the Sun. The attitude error calculation must therefore be concerned with two separate issues: keeping track of the moving solar target and accurately estimating the current spacecraft attitude.

Targeting

The default target for Inertial mode and Science mode roll axis is a quaternion rotating from the geocentric inertial frame (GCI) to the solar north reference frame (SNR), which is created by converting the following direction cosine matrix into a quaternion:

$$\mathbf{A}_{GCI}^{SNR} = \begin{bmatrix} \mathbf{X_{SNR}}^{\mathrm{T}} \\ \mathbf{Y_{SNR}}^{\mathrm{T}} \\ \mathbf{Z_{SNR}}^{\mathrm{T}} \end{bmatrix} \tag{1}$$

where $\mathbf{X_{SNR}}$, $\mathbf{Y_{SNR}}$, and $\mathbf{Z_{SNR}}$ are defined as follows:

$$\begin{aligned} \mathbf{X_{SNR}} &= \hat{\mathbf{S}}_{\mathbf{A}} \\ \mathbf{Y_{SNR}} &= \frac{\hat{\mathbf{N}} \times \mathbf{X_{SNR}}}{\|\hat{\mathbf{N}} \times \mathbf{X_{SNR}}\|} \\ \mathbf{Z_{SNR}} &= \mathbf{X_{SNR}} \times \mathbf{Y_{SNR}} \end{aligned} \tag{2}$$

where $\hat{\mathbf{S}}_{\mathbf{A}}$ refers to the solar unit vector from the spacecraft to the Sun corrected for velocity aberration, and $\hat{\mathbf{N}}$ is the unit vector in the direction of the solar north pole, which is a known constant. Therefore, the X-axis of the spacecraft is aligned with the apparent Sun center, the Z-axis is roughly aligned with the solar north pole, and the Y-axis completes the triad. Because the Sun moves at about 0.04 degrees per hour, $\hat{\mathbf{S}}_{\mathbf{A}}$ is time-varying and is calculated onboard the spacecraft. The spacecraft target, therefore, is also time varying, and takes into account the movement of the Sun.

The default target quaternion can, in addition, be modified by multiplying by a delta quaternion representing the rotation from the SNR to the target frame (TAR). All of the instrument calibration maneuvers are expressed as desired offsets from Sun center, so delta quaternions are used to command all of these maneuvers except for the guide telescope calibration maneuvers.

The Science mode pitch and yaw attitude errors come directly from the CGT processing. Two bias terms that are added to the guide telescope measurement are included in the processing. The first bias is the angular offset between the apparent Sun center and the science reference boresight (SRB). For a description of the SRB, see Reference 2. The second bias is the angular offset between the SRB and the desired calibration point. If the calibration biases are zero, adding the SRB bias to the CGT measurements effectively creates an artificial attitude error that causes the spacecraft to slew such that the SRB is pointed at the center of the Sun. If the calibration biases are not zero, adding both biases to the CGT measurements effectively creates an artificial attitude error that causes the spacecraft to slew such that the SRB is pointed off of Sun center, at the desired calibration point.

Attitude Estimation

As discussed previously, in Science mode the pitch and yaw attitude errors come directly from the CGT processing, and no attitude estimation is needed in those axes. However, to calculate the attitude errors for the Science mode roll axis and all three axes in Inertial mode, which have no valid GT measurements, the ACS needs an estimate of the spacecraft attitude. Recall that most of the instrument calibration maneuvers are performed in Inertial mode, and that these maneuvers impose a tight attitude knowledge requirement of 35/70/70 arcsec, 3σ, on the ACS. Therefore, obtaining a good attitude estimate is a high priority for the ACS. To accomplish this attitude estimation, the ACS design uses a multi-step attitude determination process.

First, the current spacecraft rate, as measured by the inertial reference unit (IRU) and corrected for gyro drift, is used to propagate the previous attitude estimate to the current time. This resulting attitude is the expected attitude of the spacecraft. Next, a six-state, linearized, Extended Kalman Filter (KF) is used to estimate the attitude errors (three states) and gyro drift bias errors (three states). This KF is similar to the one used on the Wilkinson Microwave Anisotropy Probe (WMAP).[3] The attitude errors are then used to update the expected attitude to obtain the current spacecraft attitude estimate. Finally, the gyro drift bias errors are used to update the estimate of the gyro drift. The preliminary ACS design used two star trackers (ST) as inputs into the KF, which, under nominal circumstances, would meet the ACS attitude knowledge requirement even in the event of a ST failure.

However, the SDO mission's geosynchronous orbit is a harsh radiation environment. Over the five-year mission life, charge couple devices (CCDs), such as those used in the STs, will accumulate radiation damage. With the vehicle maintaining an inertially-fixed attitude, the stars observed by each ST will remain on the same pixels for hundreds of hours, causing the spatial errors that would be averaged out if the stars moved quickly across the CCDs to persist in the ST attitude solution as an unpredictable "dark current" bias. Using worst-case estimates of the dark current bias and combining the two ST measurements in the KF, the resulting attitude solution barely meets the ACS attitude knowledge requirement. Should one of the STs fail, ACS could not meet its knowledge requirement. SDO is a single fault tolerant mission, so the ACS team had to determine a way to reduce the impact of the dark current bias term on the attitude knowledge during instrument calibration maneuvers. An attitude determination trade study was performed that provided two potential solutions to the attitude knowledge problem: IRU propagation or the addition of a digital Sun sensor (DSS).

IRU Propagation

The first option avoids additional hardware costs by maintaining the original ACS sensor suite and uses an operational work-around to perform the Inertial mode instrument calibration maneuvers. As discussed previously, the attitude knowledge problem is that the STs potentially have a dark current bias that is transparent to the ACS. In the event of a ST failure, that dark current bias could be larger than the ACS attitude knowledge requirement. The proposed solution eliminates the use of the ST from the instrument calibration maneuvers, and thus reduces the attitude knowledge error. Instead, the maneuvers would be performed by using the IRUs to propagate the vehicle attitude.

Before the calibration maneuver, the spacecraft will be in Science mode using the GT signals directly. The attitude knowledge error in pitch and yaw will essentially be zero, because the attitude errors come directly from the science instrument, and these measurements are treated as truth for the ACS. The KF will be running (potentially with only one ST for updates) to estimate the gyro drift bias errors. The KF still produces an attitude estimate, but that estimate is only used to determine the roll axis attitude error. In preparation for the instrument calibration maneuver, the ACS would switch to Inertial mode and begin using the IRU propagated attitude estimate, *i.e.* the KF would be disabled. At the time of the switch, the attitude estimate would be reinitialized with the current SNR target as calculated in Equations 1 and 2. This reinitialization of the attitude estimate would ensure that the spacecraft would not move from the SRB during the switch from Science to Inertial mode. At this point in the maneuver, the attitude knowledge errors in pitch and yaw would still be essentially zero. However, by beginning the attitude propagation during the calibration maneuvers from a calculation of SNR, ACS would eliminate all direct measurement of the vehicle attitude from the pitch and yaw knowledge calculation. The roll axis knowledge error would essentially be the same, and based on worst-case ST performance numbers, is 27.5 arcsec, 3σ, for a single ST, which meets the 35-arcsec requirement.

Next, the first step in a given calibration maneuver is commanded by multiplying the default SNR target by a delta quaternion. For example, for the first calibration point in the EVE cruciform maneuver, the delta quaternion is $\Delta Q = \begin{bmatrix} 0 & \sin(2.5/2) & 0 & \cos(2.5/2) \end{bmatrix}$. The spacecraft will begin slewing, and the attitude will be estimated using IRU propagation, which is low in noise and free of the random bias jumps that the ST can potentially see. The problem with using IRU attitude propagation is that the IRUs have a random walk bias that drifts over time. When the KF is running, that bias gets estimated and essentially eliminated, but with the KF disabled, the last estimated bias will be held by the software and not updated. Over time, the gyro drift bias will grow uncorrected until the ACS no longer meets its attitude knowledge requirement. To combat this drift during the long instrument calibration maneuvers, the spacecraft will periodically have to recalibrate the gyro drift bias. This recalibration can be accomplished by returning to the Sun and switching back to Science mode. Once in Science mode, the KF can be re-enabled and the gyro drift bias estimate updated. After the gyro biases have been updated, the ACS will switch back to Inertial mode and continue the calibration maneuver from where it left off.

The frequency with which the spacecraft must return to the Sun for recalibration depends on the characteristics of the chosen IRU. For the SDO IRU, Kearfott's TARA-1T, the drift rate bias can change by as much as 265 arcsec/hr, 3σ. See Reference 2 for a complete description of the SDO ACS hardware. So, roughly every 15 minutes, the spacecraft would need to return to Sun center to recalibrate the gyro drift rate biases. This frequency of returns can significantly increase the length of the calibration maneuvers. For instance, the time required for the EVE cruciform scan, SDO's longest maneuver, would nearly double from the nine hours allocated in the data capture budget to almost eighteen hours. In total, the longer calibration maneuvers would meet requirements, at the loss of approximately 48 hours per year of science data. Besides the increased interruption to science, the whole maneuver plan undesirably complicates the operational procedures. Science data processing is made more difficult both by the frequent interruptions to the maneuver and switching attitude knowledge back and forth between the sensor measurement/KF and by the IRU propagation of the calculated target.

Digital Sun Sensor

The second option is to increase the ACS hardware sensor complement. While a third ST was considered, it was more cost-effective to include a digital sun sensor (DSS). A DSS would be less expensive, easier to add to the baseline design, and, like the GT, this sensor would provide a direct measurement of the pitch and yaw attitude relative to the center of the Sun. Unlike the GT, a DSS could keep the Sun in its field of view during all instrument calibration maneuvers. Even with the lower accuracy of a DSS compared to a fine Sun sensor like the GT, including the additional measurements in the KF estimation would provide the required attitude knowledge accuracy in the event of a ST failure.

A DSS's typical low mass and volume would be easy to accommodate mechanically. There is room on the front face of the instrument module to mount a small optical sensor without any interference in its field of view. Further, this low power hardware could easily be accommodated within the existing power budget, current thermal radiator area and spare electrical services. However, data services required careful consideration. The custom electronics design and test equipment were already completing their engineering models; thus, the addition of an analog or digital data interface would be a large impact to the attitude control electronics (ACE) and the dynamic simulator used for hardware-in-the-loop testing. To minimize cost impact and maintain schedule, any DSS option would need to provide a Mil Std 1553 output.

Further, the flight dynamics system (FDS) had heritage tools available to provide calibration products for a DSS. With the ACS attitude determination (AD) design still in progress, flight software development had not yet begun. If the change was implemented early in the detailed design phase, the ACS, FDS and flight software team could simply complete the AD system including a DSS and minimize cost and schedule impacts to already planned work.

Finally, this addition would keep operation simple. The maneuver plan would be unchanged, even in the event of a single-sensor failure. The attitude knowledge would always meet requirements needed for science data processing, and the maneuvers could be completed within the time allocated in the data capture budget.

Attitude Determination Trade Study Results

The results of the attitude determination trade study are summarized in Table 1. The attitude knowledge for different hardware complements – one ST and IRU (baselined complement with single failure); two STs and IRU (baselined complement), two STs, one DSS and IRU (complement with addition of a DSS); and one ST, one DSS and IRU (complement with addition of a DSS and a single failure) – were calculated. The knowledge error budget is dominated by the "dark-current" bias term. Kalman filter performance and ACS sensor alignment errors were estimated using the attitude determination software, ADEAS.[4] In addition, IRU quantization effects and time-tag errors were included.

Table 1: Attitude determination trade study results.

Sensor Comple-ment	**AD Accuracy (arcsec, 3-σ)**	**Margin Against AD Requirements**	**Baseline AD**	**Addition of DSS to AD**
	Reqd: 35/70/70			
1 ST	19/47/84	82%/48%/-*17%*	Baseline AD design with a single failure – **Does not meet requirements**	
2 ST	14/40/54	155%/76%/33%	Baseline AD design	Still meets requirements in the event of a DSS failure
2 ST, 1 DSS	14/34/40	155%/106%/72%		3 sensors, new nominal performance
1 ST, 1 DSS	19/38/50	81%/84%/40%		Still meets requirements in the event of a ST failure

With the addition of a DSS, the SDO attitude determination system becomes two-for-three fault tolerant. As the above results show, if either the ST or the DSS fails, the knowledge requirements are still met with reasonable margin. Note that knowledge accuracy for the IRU propagation option is not explicitly calculated. As described above, this option will always meet the attitude knowledge requirement by both removing the ST, and therefore its dark current bias error, from the estimation and interrupting the science calibration maneuvers frequently enough to ensure the gyro drift bias error remains within allowable limits.

The SDO team chose to add a DSS to the baseline observatory design. Despite the additional money and resources required to procure and accommodate a DSS, the simple, straightforward change allows us to meet attitude knowledge and data completeness requirements easily, even in the event of a failure. By making the decision early in the detailed design phase, the mission was able to procure the DSS along with the already planned commercial components. All subsystems were able to update their designs and include the sensor in their critical design review (CDR) work.

DEVELOPMENT OF SCIENCE AND INERTIAL MODE CONTROLLERS

Developing a scheme to estimate the spacecraft attitude errors such that the ACS meets its knowledge requirement is only part of the overall design problem of performing the science instrument calibration maneuvers. Once the controller knows its attitude errors, it must still slew the spacecraft within its allotted time and hold it steady on its target. These concerns are addressed in the design of the controller itself.

Controller Description

Early in the design of the Science and Inertial mode controllers, the primary concern was to maintain tight, steady pointing on a given target, nominally the Sun. For these modes, the ACS team chose a proportional-integral-derivative (PID) controller because of its relative simplicity and its ability to remove steady state error. The controller acts to null the attitude and rate errors. Structural filters on the calculated torque were added to reduce spacecraft jitter. The controller integrator was limited to prevent wind-up in the integral term. Also included in the controllers was torque scaling logic, which maintains the desired torque direction and reaction wheel assembly (RWA) momentum redistribution logic, which acts to keep the wheels away from zero speed where static friction can increase spacecraft jitter and reduce RWA life. As mentioned previously, the Science mode pitch and yaw attitude error comes directly from the GT data processing. The rate error is measured directly by the IRUs. For Science mode roll and all of Inertial mode, the attitude error is calculated as the small-angle difference between a target attitude quaternion and the estimated attitude quaternion, as discussed above. The rate error is measured directly by the IRUs.

It was not until later in the design process that the instrument calibration maneuvers were outlined, and the overall size and duration of the required maneuvers discovered. At this point, the ACS team decided to try and adapt the original, steady pointing PID controllers to be able to accommodate the instrument calibration maneuvers, most of which consist of a large number of very small slews. In early simulations, the ACS team saw that it was taking a long time to slew the spacecraft because the primary goal of the original controller gains was to hold the spacecraft steady. This slow slewing conflicted with the science data capture budget because of the large number of slews (202) required for the EVE cruciform maneuver. The original estimate of the maneuver duration was over 19 hours, which was too long and could cause the mission to not meet its data capture budget.

Initially, we attempted to adjust the controller gains to obtain faster slews. Simulations showed that these new gains resulted in faster short slews, but, for slightly longer slews (on the order of a few degrees), the total slew and settling time increased because of large overshoots. At this point, we decided to limit the maximum spacecraft rate, which would hopefully reduce the overshoot. The rate limiting is accomplished indirectly by adding an attitude error limiter. The control torque from the controller, τ_c, is calculated as follows:

$$\tau_c = K_p\theta + K_d\dot{\theta} + K_i \int \theta \tag{3}$$

where θ is the attitude error and K_p, K_d, and K_i are the proportional, derivative and integral gains, respectively, and include multiplication by the spacecraft inertia. For the spacecraft to maintain a constant rate, the calculated control torque must be zero. Neglecting the assumed small contribution of K_i, the above equation can be rewritten as follows:

$$K_p\theta = -K_d\dot{\theta} \tag{4}$$

Equation 4 implies that for the spacecraft rate to be limited, the spacecraft attitude error must also be limited. Conversely, if the spacecraft attitude error is limited, the spacecraft will reach a constant slew rate. If the desired slew rate is known, the required attitude limit can be calculated using the known proportional and derivative gains. The ACS team decided on a maximum slew rate of 0.3 deg/sec, which, with the chosen gains, resulted in an attitude error limit of 1.7 deg. Simulations with the attitude error limiter incorporated showed acceptable reductions in the overshoot and thus the overall settling time.

At this point, the ACS team felt it had a solid controller design that could meet its twofold purpose of steady pointing and fast slewing. A representative single axis block diagram of the Science mode X-axis and Inertial mode, as designed thus far, can be seen in Figure 2.

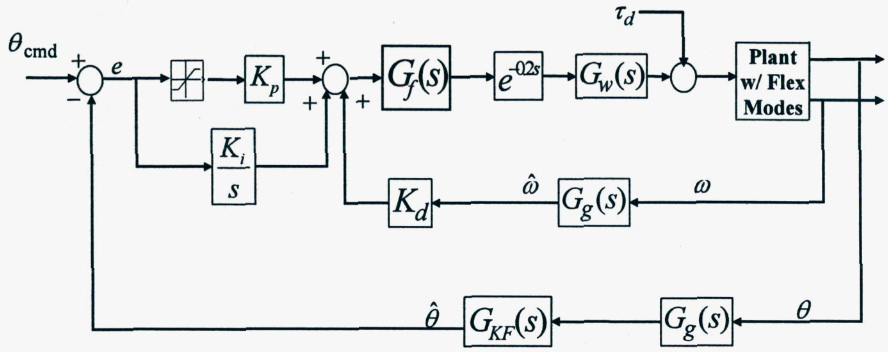

Figure 2: Inertial Mode and Science Mode X-axis Block Diagram

where $G_g(s)$ is a second-order low-pass filter representing the gyro dynamics, $G_w(s)$ is a first-order low-pass filter representing the reaction wheel dynamics, $G_{KF}(s)$ is a transfer function of 1 representing the Kalman Filter dynamics, $G_f(s)$ is a second-order elliptical filter, and $e^{-0.2s}$ represents a one-cycle delay in the 5 Hz loop. There are two differences between the above block diagram and the Science mode Y- and Z-axes. The first is that the gains, K_p, K_d, and K_i are different. The second is the replacement of $G_{KF}(s)$ with $G_{GT}(s)$, which represents the GT dynamics, but is also assumed to be a transfer function of 1.

Controller Redesign

During the guidance, navigation, and control (GNC) subsystem Critical Design Review (CDR), concern arose about the stability of the Science mode X-axis and Inertial mode controllers due to the combination of the attitude limit and the torque scaling. To address this concern, the ACS team used a combination of root locus and Routh stability techniques to analyze the controller stability.

First, we examine the stability concern itself. Neglecting the structural filter, one-cycle delay, gyro and wheel dynamics, and the plant flexible modes, the following Routh stability criterion can be calculated for the PID controller with torque scaling:

$$K_i < \frac{K_p K_d K_t}{I} \tag{5}$$

where K_t is the torque gain and I is the spacecraft inertia. K_t can range from 0 to 1 and represents the fraction of commanded torque after torque scaling vs. desired torque as calculated by the controller. For SDO, the controller gains are defined to scale with the spacecraft inertia, so the Routh stability criterion can be rewritten as follows:

$$Ik_i < \frac{I^2 k_p k_d k_t}{I} \tag{6}$$

or

$$k_i < k_p k_d k_t \tag{7}$$

where for Science mode X-axis and Inertial mode $k_i = 0.005129\,\text{Nm/kg}\cdot\text{m}^2\cdot\text{s}\cdot\text{rad}$, $k_p = 0.07192\,\text{Nm/kg}\cdot\text{m}^2\cdot\text{s}$, and $k_d = 0.467055\,\text{Nms/kg}\cdot\text{m}^2\cdot\text{rad}$. Consider the case where $k_t = 1$, *i.e.* there is no torque scaling. Using the above gains, if there is no attitude limiting, then the controller meets the above inequality and is stable. However, when the attitude is limited, it effectively reduces k_p, which creates the possibility for the Routh criterion not to be met and thus for the controller to be unstable. To avoid this potential instability, the ACS team decided to disable k_i when the attitude error is above the saturation limit of 1.7 deg, which results in the following Routh stability criterion:

$$0 < k_p k_d \tag{8}$$

which, theoretically, will always be stable. Recall, however, that the above Routh stability criterion does not include the structural filter, one-cycle delay, gyro and wheel dynamics, and plant flexible modes. To determine the effect of adding these terms, the block diagram in Figure 3 can be used to plot the Root Locus of the closed loop Inertial mode system with k_i disabled, where K_p is assumed to be the varying gain and includes the spacecraft inertia.

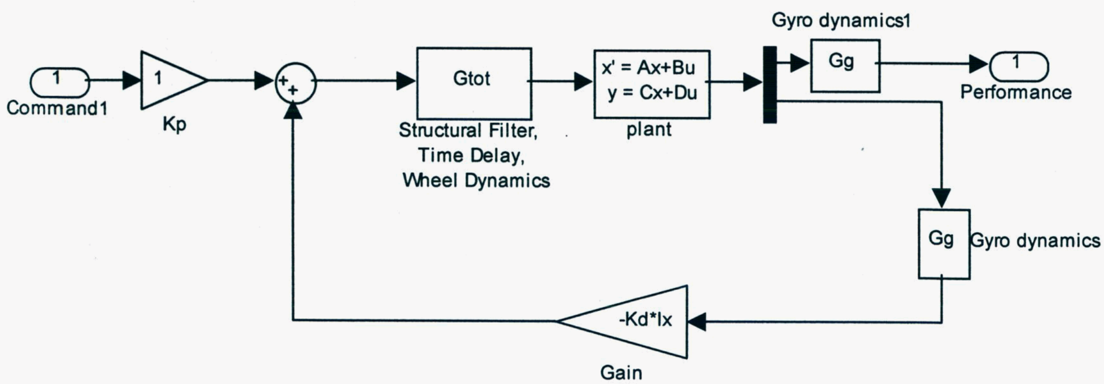

Figure 3: Block diagram used to determine the stability of the PD controller with attitude limit.

The resulting root locus plot is shown in Figure 4. As can be seen, the root locus does go into the right half plane (RHP), indicating that a certain range of K_p values can cause the system to go unstable even when integral action is disabled. Those values were found to be K_p greater than 889 Nm/rad. The nominal K_p values for Science mode X-axis and Inertial mode, including the spacecraft inertia, are $K_p = \begin{bmatrix} 124 & 202 & 171 \end{bmatrix}$ Nm/rad for the X-, Y-, and Z-axes respectively. Thus, limiting the attitude, which effectively reduces K_p, cannot cause the system to go unstable when the integral term is turned off and no torque scaling is present.

As mentioned previously, the above attitude limit analysis assumed $k_t = 1$; *i.e.* there is no torque saturation. As discussed in Reference 2, the maximum output torque per wheel of the SDO reaction wheels

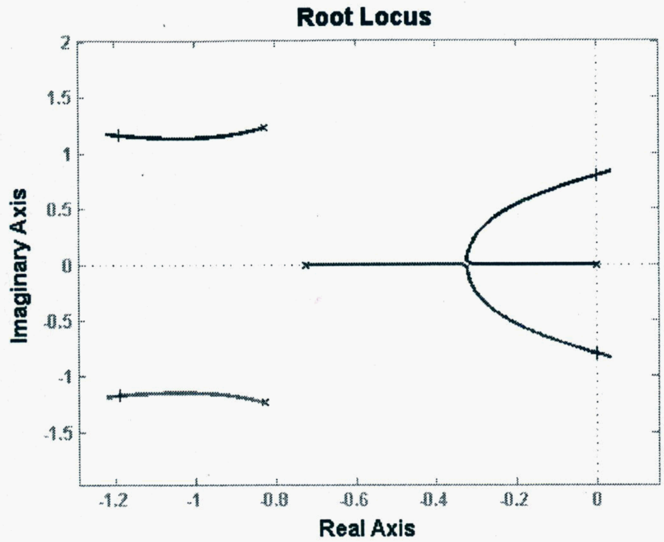

Figure 4: Root locus of Science mode X-axis and Inertial mode with no integral gain and assuming K_p can vary.

is 0.3 Nm; but, because of software limiting, the maximum torque per wheel is limited to 0.2 Nm. With the current wheel configuration, this assumption implies a minimum torque capability of 0.2 Nm for any given body axis. Using the Science mode X-axis and Inertial mode gains, this torque limit implies that an attitude slew of 0.06 deg will saturate the wheels. Referring back to Equation 7, if the integral gain is active, scaling the torque could cause an instability in the system. As mentioned earlier, we decided to disable the integral action when the attitude error is above the attitude limit; however, the 0.06-deg slew limit is smaller than the 1.7-deg attitude limit, so the possibility for instability still exists. At the same time, torque saturation by itself does not mean that the system will be unstable. As before, root locus techniques can be used to determine at what fraction of output torque over desired torque the system will go unstable. Figure 5 shows the block diagram used to plot the torque root locus, where no attitude limiting is assumed and k_t is assumed to be the gain that varies.

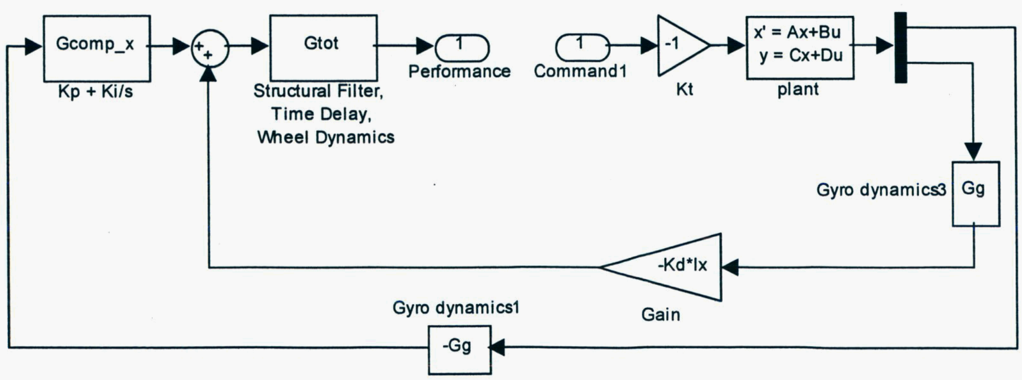

Figure 5: Block diagram used to determine the stability of the PID controller with torque saturation.

The resulting root locus plot assuming k_t can vary from 0 (no output torque) to 1 (output torque equals desired torque) can be found in Figure 6. As can be seen, part of the root locus does go into the RHP, so there are values of k_t for which the system will go unstable if the integral action is enabled. That k_t is found to be 0.17, so the system output torque must be at least 17% of the desired torque in order for the system to remain stable.

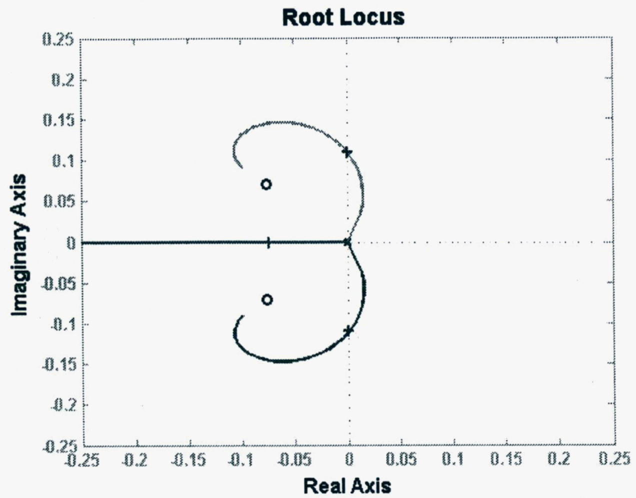

Figure 6: Root locus of Science mode X-axis and Inertial mode with integral gain and no attitude limit, and assuming k_t can vary from 0 to 1.

Knowing that the combination of attitude limiting and torque saturation could potentially cause the system to go unstable, we had to determine a method by which system stability could be ensured. The desire was to incorporate a simple method of disabling the integral action whenever system stability was in question. Several methods were investigated and a trade study performed to determine which method would be the best to incorporate. These methods and the details of the trade study will be presented in a future paper.

The method of stability assurance the ACS team decided upon was to check both attitude errors and rate errors. If either the attitude error or the rate error for any spacecraft axis is greater than a predefined limit, the input into the integrator will be set to zero. The limits were selected based on the expected system disturbance torques to ensure that when the integral torque is present, the sum of the proportional, derivative, and integral torques will be within the saturation level of the wheels. Zeroing the input into the integrator does not remove the integral term entirely; it simply keeps that term from changing. In effect, when the attitude or rate errors are outside their limits, the integral term becomes a constant disturbance torque acting on the system. The resulting PD controller cannot entirely reject that disturbance, but neither will it cause the system to go unstable. When the spacecraft attitude and rate errors are back within their limits, the integral torque is allowed to change again, and any steady state error is removed. By incorporating these attitude and rate limit checks, the ACS team can ensure the stability of the Inertial mode and Science mode controllers.

SIMULATION RESULTS

Once the controller design changes were implemented, a series of simulation runs were performed using the SDO High-Fidelity (HiFi) simulation. Each of the instrument calibration maneuvers were simulated and results checked to ensure all slews were performed within the allotted time, all steady state attitude errors return to zero at the end of each slew, etc. The following section shows several representative plots of these simulations.

Figure 7 shows the EVE Cruciform Maneuver, which is performed in Inertial mode. The left-hand plot shows the angle between the spacecraft X-axis and the apparent Sun vector (Sun vector corrected for velocity aberration). As can be seen, the spacecraft slews 2.5 degrees off the Sun and steps back towards and through Sun center in 180 arcsec steps until it reaches -2.5 deg. The procedure is then repeated for the other axis. The right-hand plot shows a zoom-in of the attitude errors. As can be seen, there is no steady state error at the end of any slews. All slew times are within their requirements.

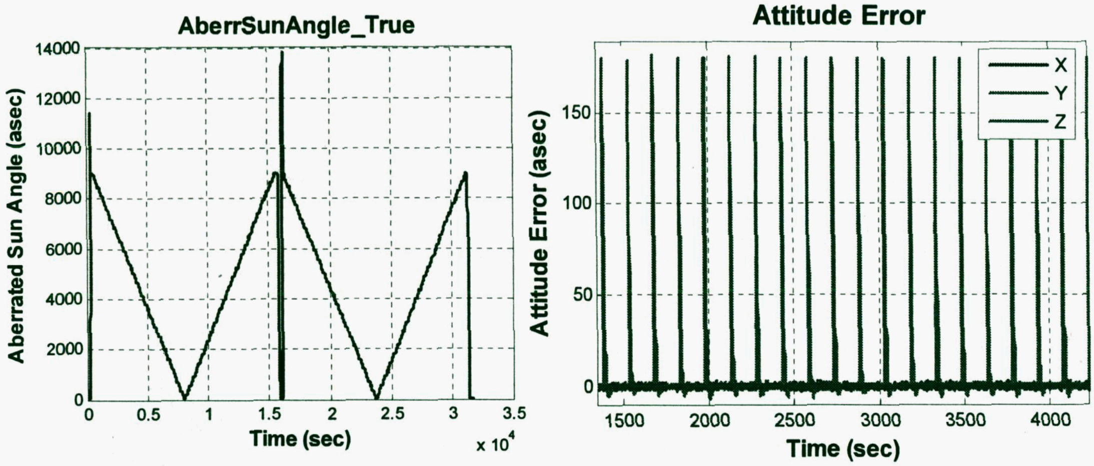

Figure 7: EVE Cruciform Maneuver. Left plot shows angle between the spacecraft X-axis and apparent Sun vector. Right plot shows zoom-in of attitude errors

Figure 8 shows the HMI/AIA Roll Maneuver. Recall that this maneuver is performed in Science mode. The left-hand plot shows the attitude errors, while the right-hand plot shows a zoom-in of those errors. As can be seen in the figure, the spacecraft is rolling about the X-axis in increments of 22.5 deg. Between each calibration point, steady state attitude errors go to zero. All slews are performed within their allotted times.

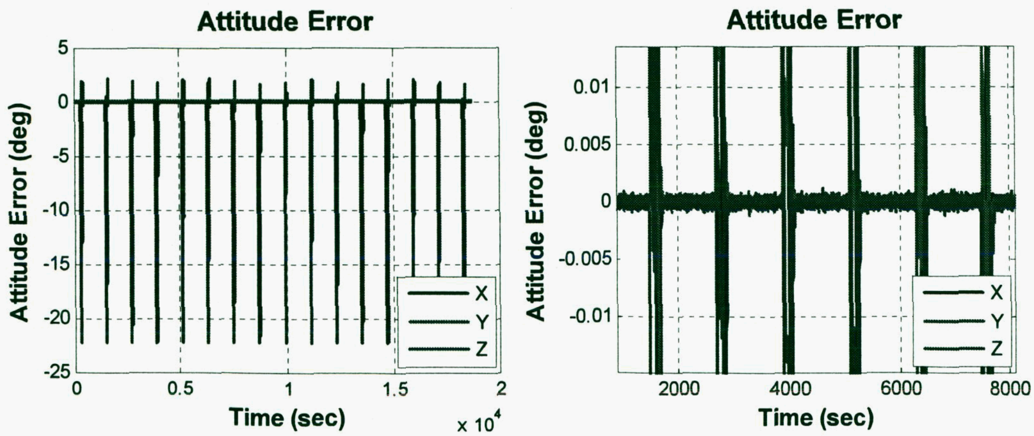

Figure 8: HMI/AIA Roll Maneuver. Left plot shows attitude errors. Right plot shows zoom-in of attitude errors

Figure 9 shows the Short Form Guide Telescope Maneuver. Recall that this maneuver is performed in Science mode using the GT calibration biases to induce slews. The left-hand plot shows the angle between the spacecraft X-axis and the apparent Sun vector. As can be seen, the spacecraft slews about 30 arcsec off the Sun and steps back towards and through Sun center in 5 arcsec steps until it reaches -30 arcsec. This procedure is then repeated for the other axis. The right-hand plot shows the attitude errors. The large spikes indicate the 30-arcsec slews. The small spike indicate the 5-arcsec steps. As can be seen, all attitude errors go to zero after each slew. Furthermore, all slew times are within their requirements.

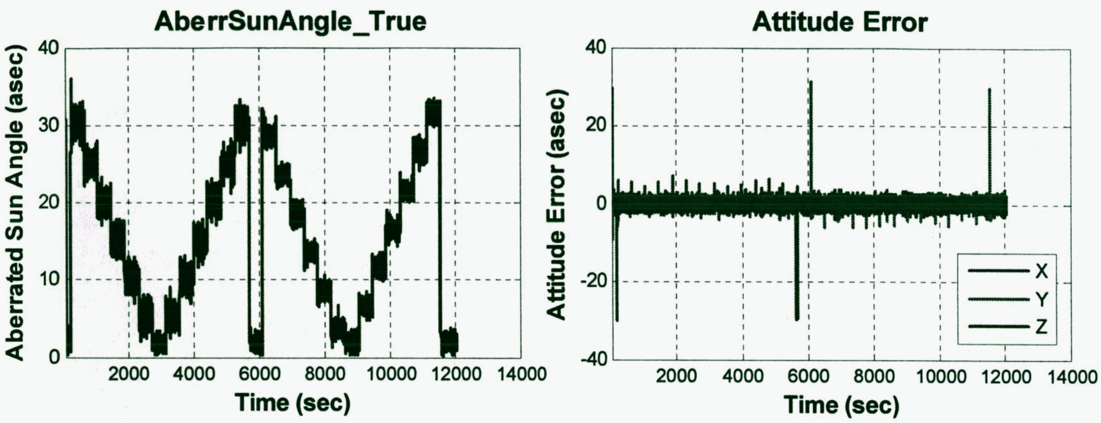

Figure 9: Guide Telescope Short Form Maneuver. Left plot shows angle between the spacecraft X-axis and apparent Sun vector. Right plot shows zoom-in of attitude errors

CONCLUSION

The science instrument calibration maneuvers are very important to maintaining the integrity of the collected science data. To ensure the instruments can collect good data during the maneuvers, the ACS has to meet tight pointing, knowledge, and slew time requirements. By incorporating a DSS in the ACS sensor suite, the ACS could ensure they meet the knowledge requirement even in the event of a sensor failure. The Science and Inertial mode controllers were designed to both slew quickly and hold steady in order to meet the pointing and slew time requirements. A scheme was developed to ensure controller stability during slews. The simulation results show that the ACS can meet their requirements during science instrument calibration maneuvers.

REFERENCES

[1] Solar Dynamics Observatory Home Page. http://sdo.gsfc.nasa.gov. responsible official: Liz Citrin, Goddard Space Flight Center, August 2005.

[2] Scott R. Starin, Kristin L. Bourkland, Kuo-Chia Liu, Paul A. C. Mason, Melissa F. Vess, Stephen F. Andrews, and Wendy M. Morgenstern. Attitude Control System Design for the Solar Dynamics Observatory. *Flight Mechanics Symposium*, NASA CP–2005-XXXXXX, 2005.

[3] Stephen F. Andrews and James R. O'Donnell, Jr. MAP Attitude Control System Design and Flight Performance. *AIAA Paper 2002-4580*, 2002.

[4] F. Markley, E. Seidewitz, and M. Nicholson. A General Model for Attitude Determination Error Analysis. *Flight Mechanics/Estimation Theory Symposium*, Goddard Space Flight Center, May 10-11, 1988, pp. 3-25.